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N73-16496
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IN HYPERSONIC ENGINE TESTING**

by George E. Glaue

Lewis Research Center

Cleveland, Ohio 44135

1. Report No. NASA TM X-2715		2. Government Accession No.		3. Recipient's Catalog No.	
4. Title and Subtitle A COOLED-GAS PYROMETER FOR USE IN HYPERSONIC ENGINE TESTING				5. Report Date February 1973	
				6. Performing Organization Code	
7. Author(s) George E. Glawe				8. Performing Organization Report No. E-7214	
9. Performing Organization Name and Address Lewis Research Center National Aeronautics and Space Administration Cleveland, Ohio 44135				10. Work Unit No. 501-24	
				11. Contract or Grant No.	
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546				13. Type of Report and Period Covered Technical Memorandum	
				14. Sponsoring Agency Code	
15. Supplementary Notes					
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17. Key Words (Suggested by Author(s)) Instrumentation Hypersonic flow measurement Temperature measurement Fluid flow				18. Distribution Statement Unclassified - unlimited	
19. Security Classif. (of this report) Unclassified		20. Security Classif. (of this page) Unclassified		21. No. of Pages 17	
				22. Price* \$3.00	

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SUMMARY

A cooled-gas pyrometer designed for application in a hypersonic research engine program was fabricated and tested. Design and operational considerations and calibration data are presented. The probe was tested in a rocket-engine exhaust stream operating at Mach 2 and 2300 K. Test temperature measurements agreed to within 2 percent with a radiation shielded Ir40Rh/Ir thermocouple probe.

INTRODUCTION

The engineering development of propulsion systems for hypersonic flight vehicles involves studies of components and testing of the complete engine. Measurements involve application of various techniques for determining enthalpy, temperature, pressure, gas composition, and flow rates through the test units. Gas temperature measurements are usually beyond the range of commonly used thermocouples so that other types of probes must be used to obtain temperature or enthalpy. The cooled-gas pyrometer, initially introduced in reference 1, has previously been used in a supersonic combustion study (ref. 2) and has been proposed (ref. 3) for possible application in other areas of hypersonic propulsion work. This report presents the design and testing of a cooled-gas pyrometer for measurements in the exhaust stream of a hypersonic engine. Probe tests ranged to 2300 K in a Mach 2 stream. The primary quantity obtained with the cooled-gas pyrometer is enthalpy or its related temperature; however, the probe may also lend itself to total pressure measurement and gas sampling.

DESIGN CONSIDERATIONS

General Considerations

The cooled-gas pyrometer utilizes the controlled cooling of a continuously aspirated sample of the gas whose temperature is to be measured. This is accomplished by

drawing the hot gas through a tube with cooled walls. The loss of energy of the gas is reflected by a drop in gas temperature. This temperature drop is a function of the flow in the tube, gas properties, tube geometry, and tube wall temperature. Therefore, if the temperature of the cooled gas is measured and if the energy loss by cooling can be calculated, the free stream total temperature of the gas entering the probe can be determined. The correlation equation relating total temperature T_0 to the probe internal gas temperature T_i , tube wall temperature T_w , and a Prandtl-Reynolds number relation, is in a general form (ref. 1):

$$\ln\left(\frac{T_0 - T_w}{T_i - T_w}\right) = CPr^{-2/3}Re^{-a}$$

where the constant C and the exponent a are determined by calibration. The actual "working" correlation equation for the probe where the Prandtl and Reynolds number terms are evaluated from probe measurements will be shown later.

The present probe design was based on the theoretical and experimental criteria presented in references 1, 4, and 5. The presently proposed application, however, imposes some new considerations, which dictate important changes from the original design concept.

Figure 1 is a schematic diagram of the original probe design. (Symbols and nomenclature are included in the sample computation in the appendix.) The free stream gas at station 0 is drawn into the probe at station 1 and is cooled while passing through the inside tube whose wall temperature is kept essentially constant by a water jacket. At station 2, the gas has cooled sufficiently to obtain a measurement T_2 with a thermocouple located at the center of the tube. In the original design (ref. 1) a tapered body, with a square section attached to the thermocouple insert (fig. 1) acted as a flow nozzle whose minimum area was the flow passage between the square body in the right circular cylinder tube at station 3. The pressure downstream of this section was kept sufficiently low (and monitored by tap p_4) to maintain sonic flow at station 3. Reference 1 develops the correlation that is used to obtain gas total temperature T_0 , from measurements of inlet pressure P_0 , total temperature T_2 , approximate wall temperature T_w , and gas property values. The pressure P_0 is free stream total pressure for subsonic flow, or the total pressure behind a normal shock for supersonic flow.

The thermocouple insert configuration of the original design was a convenient and flexible method for setting the distance between stations 1 and 2. For a particular application, where a free choice can be made of the tube diameter d and the axial distance L , the insert can be positioned so that the temperature at station 2 is sufficiently low to use a Chromel-Alumel thermocouple on an Inconel nozzle body.

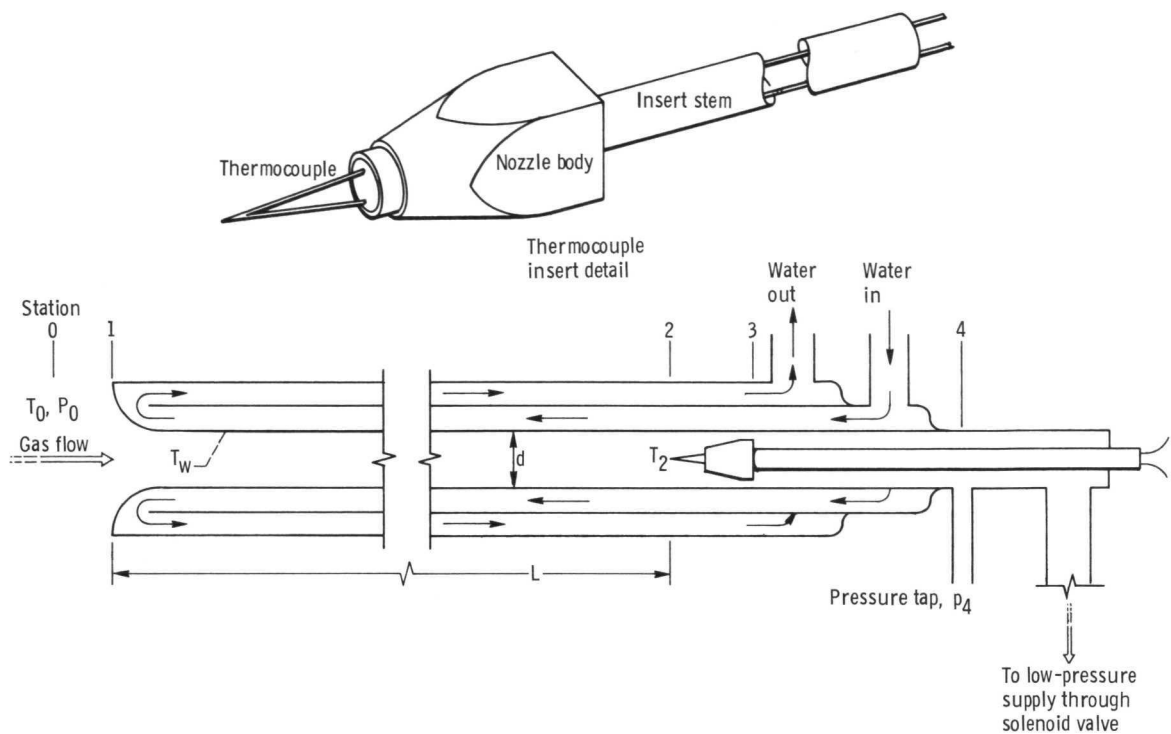


Figure 1. - Original design, cooled-gas pyrometer with thermocouple insert detail.

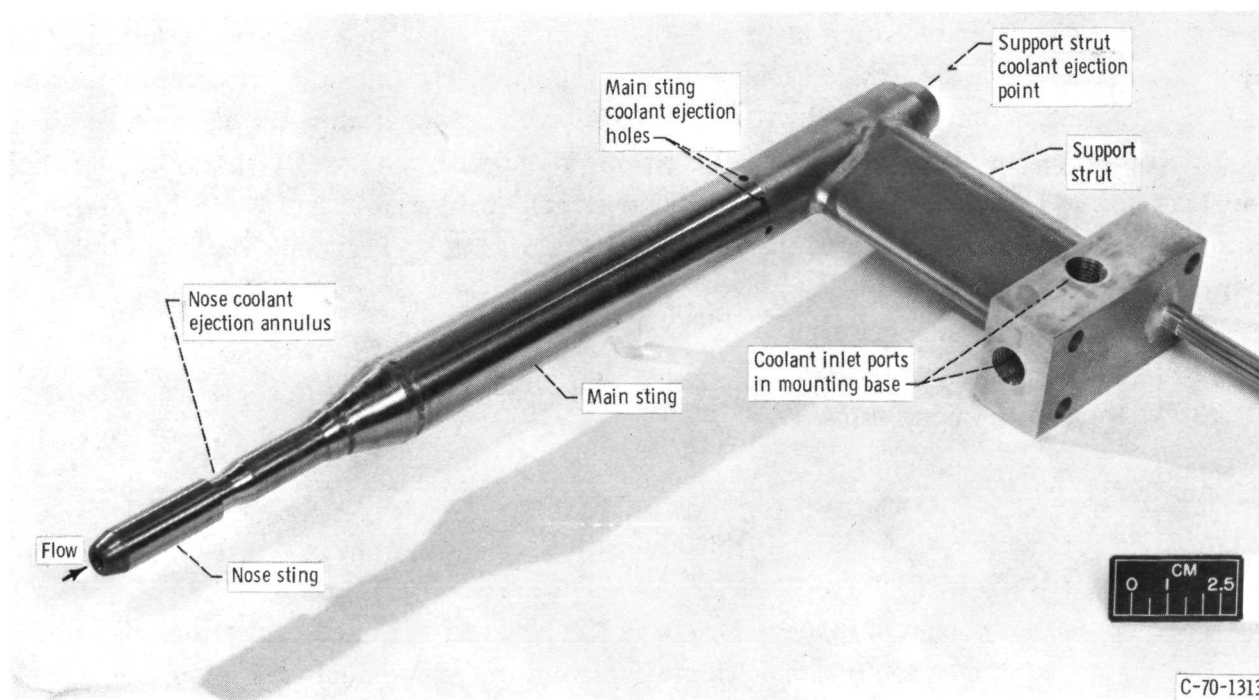


Figure 2. - Present design, cooled-gas pyrometer.

In the present application, physical requirements reduced the L/d ratio to a point where the temperature at station 2 precluded the use of the original type thermocouple insert. Consequently, the new probe, designed for hypersonic engine testing (figs. 2 and 3) incorporates a cooled Venturi section with a noble metal thermocouple at station 2, capable of operating at a temperature suitably higher than the old insert assembly.

Furthermore, because of the L/d consideration and space limitations, the inside tube diameter d of the new probe was made smaller than in the original design. This, along with operation at high temperature and low pressure, would place the probe in a low Reynolds number range which, was likely to involve operating in a region of transition from laminar to turbulent flow. Operating in this region increases the uncertainty in correlating the measured quantities with stream temperature and reduces the ability of the probe to cool the ingested gas to an acceptable T_2 limit. In order to counter these effects, a boundary layer trip was built into the probe about one diameter downstream of the inlet, and the internal surface of the sampling tube was roughened to promote increased cooling between stations 1 and 2.

Heat Load

Maximum heat load to the probe structure was calculated from the engine internal flow conditions anticipated for a Mach 6 flight condition. Heat loads were calculated for the probe stationed in the exhaust nozzle where the exhaust gas is flowing at Mach 2 and 2830 K. The maximum heat load will be imposed at the stagnation point of the 1.3-millimeter radius inlet lip. The heat flux at this point will be about 1000 W/cm^2 . The heat load on the leading edge of the support strut was estimated at 610 W/cm^2 . The total heat input on all surfaces of the probe is 80 kilowatts. This heat load is distributed over the probe structure as follows:

- (1) 42 Percent on support strut
- (2) 45 Percent on main sting
- (3) 13 Percent on nose sting

Probe Cooling

The engine test program made allowances for ejecting the used probe coolant (water) into the engine exhaust-gas stream. This simplified the probe cooling system design since internal flow-return passages would not be necessary. The probe cooling circuit was designed with three ejection points (figs. 2 and 3) having the following flow patterns. All of the coolant enters the base of the support strut, flows through the strut and ejects

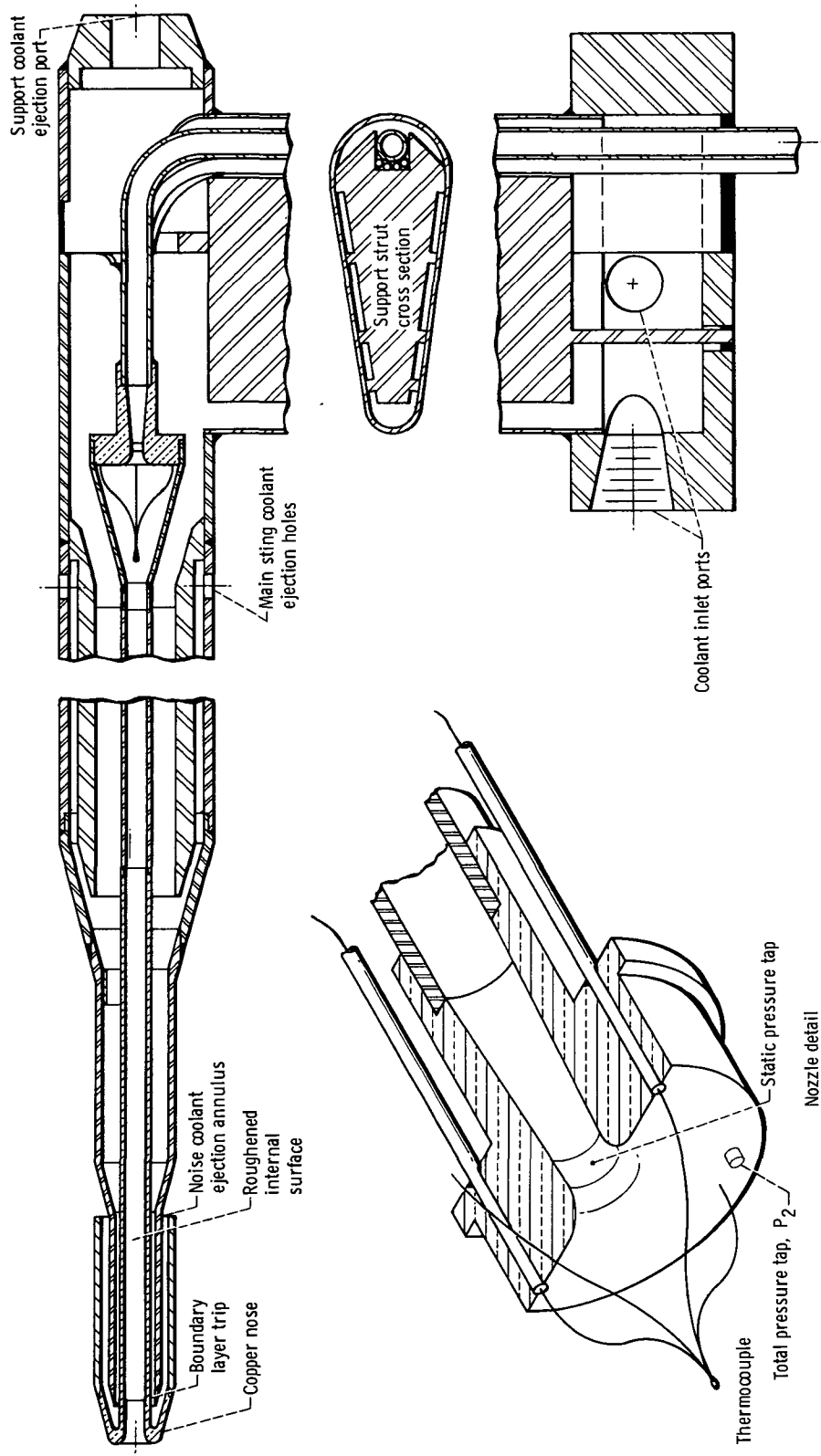


Figure 3. - Present design, cooled-gas pyrometer details.

some of the coolant at the rear of the main sting. The remaining coolant flows internally to the forward end of the main 2.5-centimeter-diameter sting and then internally reverses back to the second ejection point. Remaining internal flow continues forward to the nose, reverses, and is ejected downstream from the rear of the nose sting region.

After the probe was fabricated, a flow test was made to measure the coolant distribution to the three zones. Coolant was run through the probe and the pressure drop measured from inlet to ejection points. With flow issuing freely from all three ejection points the coolant was captured and the flow rates measured. The coolant flow discharge was 40, 42, and 18 percent for the support, main sting, and nose sting regions, respectively. Cooling curves of coolant flow rate as a function of pressure drop were thus generated. By equating the coolant flow rates with the previously calculated heat load data, it was determined that with a coolant supply pressure at $7 \times 10^5 \text{ N/m}^2$ above ejection pressure, the total coolant flow would be $1.6 \times 10^{-3} \text{ m}^3/\text{sec}$ and the temperature of the coolant at the nose ejection point would be 25 K above the coolant inlet temperature.

OPERATIONAL CONSIDERATIONS

Pressure Measurements

Two requirements for the probe operation are that a measurement of the total pressure P_0 at the probe inlet be obtained, since it is one of the probe correlation parameters, and that the Venturi operates under sonic flow conditions. Measurements of the probe inlet pressure P_0 can be made by the following:

- (1) An independent measurement with a Pitot tube.
- (2) Shutting off the flow through the cooled-gas probe, thus operating it as a total pressure probe.
- (3) Relating P_2 to P_0 , where P_2 is the pressure measured internally on the nozzle face (fig. 3) at station 2 (while the probe is passing flow).

The relation between P_0 and P_2 for the present probe design was experimentally determined over the pressure range of P_0 from 0.8×10^5 to $4.5 \times 10^5 \text{ N/m}^2$ and found to be

$$P_0 = (1.08 \pm 0.01)P_2$$

Therefore, P_2 can be used to determine P_0 to within 1 percent. The reason for P_2 reading low, is attributed to pressure drop between stations 1 and 2 and to pressure tap location on the face of the Venturi.

Sonic operation of the Venturi can be monitored by measuring the pressure drop across the Venturi, as measured by P_2 and a static tap in the throat at station 3, or a pressure tap in the passage downstream of the Venturi. This relation should be obtained prior to application by calibration with a flow-measuring system in series with the probe.

Temperature Measurements

The probe correlation function involves the gas temperature T_2 at probe station 2 (fig. 1) and the tube wall temperature T_w . The internal gas temperature measurement at station 2 is made with a Pt13Rh/Pt thermocouple. Approximate tube wall temperature is estimated from measurement of coolant inlet temperature and calculated coolant temperature rise in the probe.

If this probe is used in a program where unscheduled, transient, over-temperature peaks may occur, the thermocouple should be protected from burnout by a reverse flow of inert gas (except when taking a reading) or by a quick-acting valve in the probe suction line which shuts off when the thermocouple reaches a value approaching its melting point.

Calibration Stability

The probe calibration can change if:

- (1) Heat transfer characteristics of the cooling tube are altered by stream erosion or particle deposition.
- (2) Stream particles deposit in Venturi throat, thus changing sonic flow area.

These factors may be alleviated by letting the probe ingest gas only when a reading is to be taken. At other times, the ideal situation would be to backflow dry nitrogen or argon. Stability of operating during a series of application runs may be checked by repeating a point which had been previously established.

One method of checking for a change in throat area is to run a room temperature calibration of the probe nozzle by a mass-flow measuring station in series with the probe flow, and then rechecking after some period of use.

PROBE CALIBRATION

The probe was calibrated in the high temperature tunnel described in reference 4. Data were obtained in air at 330 K and in the exhaust gas from the combustion of gasoline

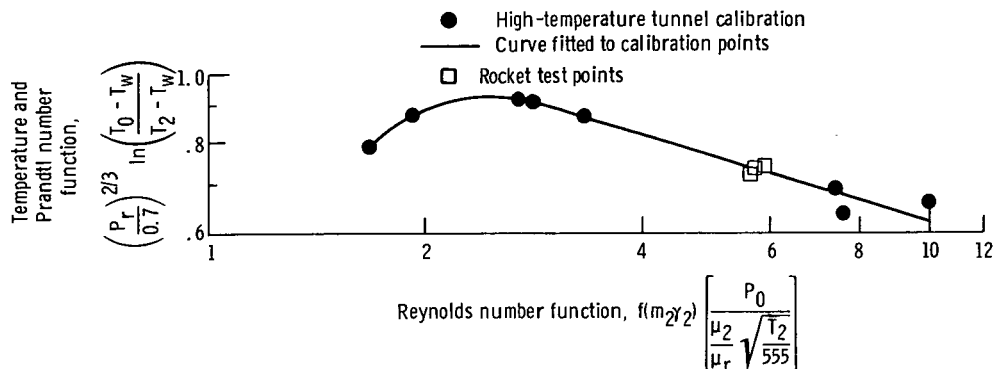


Figure 4. - Cooled-gas pyrometer correlation curve.

and air operating in the temperature range of 1000 to 1800 K. The Mach number range was 0.23 to 0.57 and total pressure ranged from 0.8×10^5 to 1.4×10^5 N/m². Stream total temperature was obtained with a shielded, aspirated Pt13Rh/Pt thermocouple (ref. 4). Gas properties at temperature T_2 were obtained from reference 6. The wall temperature T_w was taken as the average of the coolant temperature measured at the inlet and at the third ejection point. The use of this estimate of T_w was validated during the original probe development work and is based on the assumption of a small temperature drop across the cooling tube wall and the relative insensitivity of small variations in T_w in the correlation term. The probe correlation curve is shown in figure 4, where the abscissa is a function of molecular weight, specific heat ratio, total pressure, viscosity, and T_2 , and the ordinate is a function of T_0 , T_w , T_2 , and Prandtl number. The calibration covered the same range (in terms of the values of the abscissa) as is scheduled in the proposed engine tests. A sample computation, applying the correlation curve to a proposed engine test point and defining correlation terms, is presented in the appendix.

TEST IN ROCKET EXHAUST

In order to expose the probe to a higher heat load and a higher temperature environment than the high-temperature tunnel (ref. 4) could provide, a test was set up to run the probe in a nitrogen-diluted, hydrogen-oxygen rocket exhaust gas. Nitrogen dilution was necessary to lower the stoichiometric temperature to 2300 K so that a thermocouple probe could be used as a comparative measuring device. The rocket operated at a nominal chamber pressure of 7×10^5 N/m² and had a Mach 2 nozzle. The cooled-gas pyrometer and the thermocouple probe were attached to a common frame which moved through the action of a two-position pneumatic actuator system. The thermocouple probe had an Ir40Rh/Ir thermocouple element surrounded by two iridium radiation shields. Both

probes were supplied with $8 \times 10^5 \text{ N/m}^2$ coolant water. As an added precaution, the cooled-gas aspiration line, external of the probe, was equipped with a solenoid-operated valve (fig. 1) which remained closed until steady-state nitrogen-diluted running was obtained. This was to serve two purposes:

(1) With the valve closed, the pressure tap P_2 on the face of the flow nozzle at station 2 (fig. 3) will indicate Pitot pressure P_0 . (With the valve open and gas flowing, $P_2 < P_0$.)

(2) With the valve closed at rocket startup, the cooled-gas thermocouple T_2 should be protected from an overtemperature excursion.

Figure 5 presents the data record from a typical rocket run indicating the temperature and pressure traces for the cooled-gas pyrometer which was in the stream during engine startup and had $6\frac{3}{4}$ seconds total exposure. Note the drop in pressure from the value of P_0 to P_2 when the valve was opened. Also note the magnitude of the initial temperature excursion on T_2 while the valve was closed. This occurred because the trapped column of gas between the probe inlet and the solenoid valve station was of sufficient volume to be compressed at startup and allowed some initial ingestion of the exhaust gas to the thermocouple station. For this run, the Pitot pressure ratio P_0/P_2 was 1.075. The cooled-gas thermocouple indication $T_{2, \text{ind}}$ was 1240 K. Calculation of total temperature T_0 using the correlation curve (fig. 4) yielded a value of 2290 K. The thermocouple probe for this run yielded a total temperature T_0 of 2240 K. Post

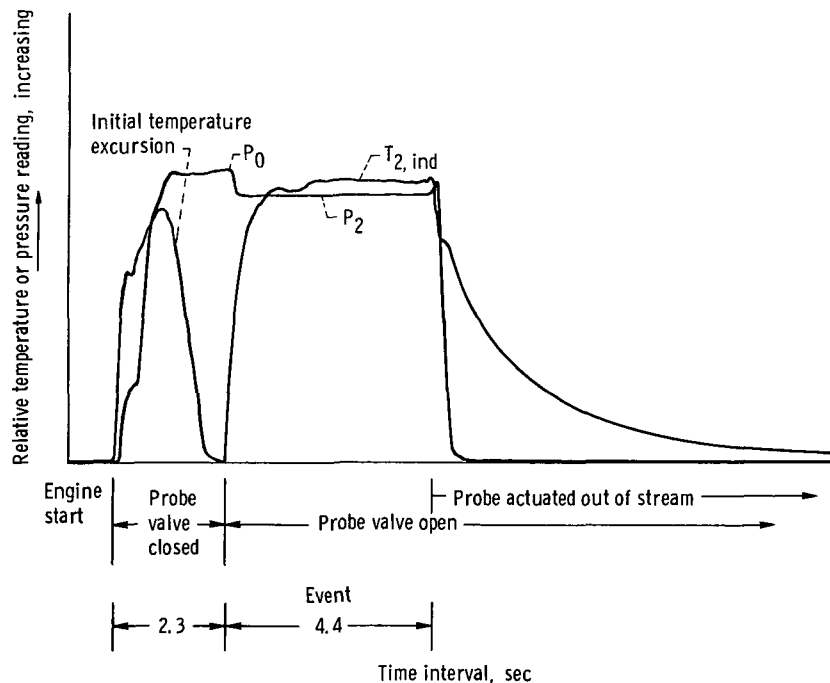


Figure 5. - Probe temperature and pressure traces from typical test in rocket engine exhaust stream.

run examination of the cooled-gas probe showed no metal discoloration at any point (which would indicate hot spots) but the thermocouple probe showed a discoloration region in a stagnation area at a 90° bend point.

Three runs were made in the rocket through a temperature range of 2100 to 2300 K, where a comparison was made with the thermocouple probe. The agreement in total temperature between the two instruments was between 1.5 and 2 percent. These data points for the rocket runs are plotted on the correlation curve of figure 4.

CONCLUDING REMARKS

A cooled-gas pyrometer was successfully tested in a high-temperature supersonic gas stream (to Mach 2 and 2300 K) which simulated conditions expected in a hypersonic propulsion engine test program.

Test temperatures obtained with the probe, using previously established calibration data, agreed with an independent temperature measurement to within 2 percent.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, November 14, 1972,
501-24.

APPENDIX - SAMPLE COMPUTATION

The hypersonic research engine test conditions are as follows:

Free stream Mach number, M_0	6
Free stream total pressure, $P_{T,0}$, N/m^2	64×10^5
Mach number at combustor exit, M_{exit}	1.5
Ratio of specific heats at combustor exit, γ_{exit}	1.16
Molecular weight at combustor exit, m	23
Total pressure at combustor exit, $P_{T,exit}$, N/m^2	5.1×10^5
Total temperature at combustor exit, T_0 , K	≈ 2830

The probe measurements in engine exhaust are as follows:

Internally measured total pressure at probe nozzle face (station 2), P_2 , N/m^2	3.29×10^5
Pitot pressure independently measured or obtained from P_2 , $P_0 \approx 1.08 P_2$, N/m^2 (atm)	3.58×10^5 (3.54)
Coolant inlet temperature measurement, T_c , K	300
Probe inner wall temperature based on T_c and estimate of average coolant temperature rise, T_w , K	≈ 312
Indicated thermocouple temperature at probe station 2, $T_{2,ind}$, K	1410

The indicated thermocouple temperature should be corrected for recovery and radiation error to obtain true temperature T_2 at probe station 2. From information in references 7, 8, and 9 along with some unpublished experimental data, the following relations can be used as an approximation of the recovery and radiation error:

$$\text{Recovery error} \approx 0.002 T_{2,ind} \quad (1)$$

$$\text{Radiation error} \approx \frac{4.5 \epsilon}{\sqrt{P_0}} \left(\frac{T_{2,ind}}{555} \right)^{3.82} \quad (2)$$

where

ϵ total hemispherical emissivity of thermocouple wire

P_0 Pitot pressure, atm

$T_{2,ind}$ thermocouple indication, K

Equations (1) and (2) may now be solved by using the appropriate preceding values along with the value of ϵ from reference 9, where $\epsilon = 0.19$. Thus,

$$\text{Recovery error} \approx 0.002 \times 1410 \text{ K} \approx 3 \text{ K}$$

$$\text{Radiation error} \approx \frac{4.5(0.19)(2.54)^{3.82}}{\sqrt{3.54}} \approx 16 \text{ K}$$

Since

$$T_2 \approx T_{2, \text{ind}} + \text{Radiation correction} + \text{Recovery correction}$$

$$T_2 \approx 1410 + 16 + 3$$

$$T_2 \approx 1430 \text{ K}$$

The property values of gas in probe evaluated at T_2 are as follows:

$$\text{Ratio of specific heats, } \gamma_2 \approx 1.32$$

$$\text{Prandtl number, } Pr_2 \approx 0.7$$

$$\text{Molecular weight, } m_2 \approx 26$$

$$\text{Viscosity, } \mu_2 \approx 5.21 \times 10^{-4} \text{ g/(cm)(sec)}$$

The abscissa for the correlation curve, figure 4, is given as

$$f(m_2 \gamma_2) \left(\frac{P_0}{\frac{\mu_2}{\mu_r} \sqrt{\frac{T_2}{555}}} \right) \quad (3)$$

where P_0 is in atmospheres, T_2 is in K, and

$$f(m_2 \gamma_2) = \left[\sqrt{m_2 \gamma_2} \left(\frac{2}{\gamma_2 + 1} \right)^{(\gamma_2 + 1)/(2\gamma_2 - 2)} \right] \quad (4)$$

Substituting the values of m_2 and γ_2 in equation (4) yields:

$$f(m_2\gamma_2) = 3.40$$

$$P_0 = 3.58 \times 10^5 \text{ N/m}^2 = 3.54 \text{ atm}$$

The viscosity function appears as a dimensionless ratio of μ_2 and μ_r . For the curve shown in figure 4, μ_r (which is arbitrarily chosen) is the viscosity of air at 555 K

μ_r reference viscosity, air at 555 K = $2.87 \times 10^{-4} \text{ g/(cm)(sec)}$

μ_2 viscosity of gas at $T_2 \approx 5.21 \times 10^{-4} \text{ g/(cm)(sec)}$

Therefore,

$$\frac{\mu_2}{\mu_r} = 1.8$$

$$\sqrt{\frac{T_2}{555}} = \sqrt{2.57} = 1.6$$

Substituting the values in equation (3) yields

$$f(m_2\gamma_2) \left(\frac{P_0}{\frac{\mu_2}{\mu_r} \sqrt{\frac{T_2}{555}}} \right) = \frac{3.4 \times 3.54}{1.8 \times 1.6} = 4.17$$

The ordinate value can now be obtained from the correlation curve of figure 4, or, since the data point falls on the linear portion of the curve, the ordinate value may be expressed as

$$1.25 \left[f(m_2\gamma_2) \left(\frac{P_0}{\frac{\mu_2}{\mu_r} \sqrt{\frac{T_2}{555}}} \right) \right]^{-0.3} = 1.25(4.17)^{-0.3} = 0.81$$

Therefore,

$$\left(\frac{\text{Pr}_2}{0.7}\right)^{2/3} \ln\left(\frac{T_0 - T_w}{T_2 - T_w}\right) = 0.81$$

and with $\text{Pr}_2 \approx 0.7$

$$\ln\left(\frac{T_0 - T_w}{T_2 - T_w}\right) = 0.81$$

or

$$\frac{T_0 - T_w}{T_2 - T_w} = 2.25$$

Thus,

$$\begin{aligned} T_0 &= 2.25(T_2 - T_w) + T_w \\ &= 2.25(1430 - 312) + 312 \\ &= 2827 \text{ K} \end{aligned}$$

The logarithmic temperature function in the correlation equation (also included in the ordinate value of fig. 4) is applicable to the internal-heat-transfer process in which the gas is in equilibrium and the specific heat is essentially constant. However, if the specific heat of the gas changes appreciably during the process of cooling in the probe it is necessary to treat the process on the basis of enthalpy differences instead of temperature differences. Such is the case, for example, when the measurement is made in a dissociated gas.

If the enthalpy of the gas at station 2 can be determined from knowledge of its composition and temperature, and if the enthalpy of the gas at wall temperature T_w is essentially constant along the tube wall, the terms in the logarithmic function can be replaced by enthalpy terms so that the correlation function becomes

$$\ln\left(\frac{H_0 - H_w}{H_2 - H_w}\right)$$

where

H_0 stagnation enthalpy

H_2 enthalpy of gas at T_2

H_w enthalpy of gas at tube wall temperature T_w

Thus, the instrument is used to determine total stream enthalpy, and the total temperature may then be determined from enthalpy tables.

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